



Electric Propulsion for Low Earth Orbit Communication Satellites

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Electric Propulsion For Low Earth Orbit Communication Satellites

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Electric propulsion was evaluated for orbit insertion, satellite positioning and de-orbit applications on big (hundreds of kilograms) and little (tens of kilograms) low earth orbit communication satellite constellations. A simple, constant circumferential thrusting method was used. This technique eliminates the complex guidance and control required when shading of the solar arrays must be considered. Power for propulsion was assumed to come from the existing payload power. Since the low masses of these satellites enable multiple spacecraft per launch, the ability to add spacecraft to a given launch was used as a figure of merit. When compared to chemical propulsion ammonia resistojets, ion, Hall, and pulsed plasma thrusters allowed an additional spacecraft per launch. Typical orbit insertion and de-orbit times were found to range from a few days to a few months.

INTRODUCTION

Many new, low earth orbit (LEO) communication satellite systems are being planned or put into service.¹ These LEO satellites can be defined as "Little LEOS" or "Big LEOS." LEOS stands for Low Earth Orbit Spacecraft. In general, the Little LEOS are relatively small satellites of tens of kilograms and provide non-voice messaging services in a store-and-dump method. The Big LEOS are larger satellites of hundreds of kilograms or thousands of kilograms that provide either global hand held telephone, fax, and data services or high-capacity data links for computer and video communications. The main impetus for these lower altitude satellites results from the reduced delay time of the Big LEOS when compared to geostationary satellites, and the reduced cost of spacecraft and launch services for the Little LEOS. The lower altitudes of these satellites necessitate many more satellites as opposed to the only three or four geostationary satellites, required for global coverage.

With the introduction of the geostationary satellites "Gals" and the GE 7000 series, the use of electric propulsion has begun on communication satellites.² The Gals satellite uses a Hall effect thruster and the GE 7000 series the hydrazine arcjet. Use of electric propulsion for part of the delivery of the geostationary spacecraft in addition to stationkeeping has been suggested by many authors^{3,4,5} and is being offered to users to increase their payload mass.⁶ This delivery of more payload mass in a timely fashion (1 to 3 months) is made possible by using the ever growing payload power associated with geostationary communication satellite payloads for the electric propulsion (EP) orbit insertion.⁷ Many proposed

LEO satellite systems have relatively high power payloads,⁸ which are not in use during satellite delivery and disposal and could be effectively used by an electric propulsion system to increase payload mass or reduce launch mass.

In the study described in this paper an assessment of the benefits of advanced EP for "generic" Big LEOS and Little LEOS constellations is made. The performance advantages were determined in terms of increased number of satellites per launch vehicle. These sample missions use available information on launch vehicles and sample satellite constellations to create the generic scenarios.^{2,9,10}

MISSION ANALYSIS, OPTIONS AND ASSUMPTIONS

Several mission tools were used in these analyses to provide low thrust trajectory, atmospheric drag, earth oblateness and shadow modeling. The numerical optimization program Solar Electric Propulsion Steering Program for Optimal Trajectory (SESPOT) was used for determining optimal solar electric propulsion starting orbits and optimal steering for constant and shaded thrusting orbits.¹¹ Cases with SESPOT were run which showed, that for a continuously operating electric propulsion system, a low circular EP starting orbit is near optimal for the launch vehicles considered herein. The numerical orbit integration program Systems Evaluation of Orbit Raising (SEOR), was used to test the use of circumferential steering.¹² Finally, the routine, Thrusting Orbiter with Atmospheric Drag (TOAD) was used to assess the impact of atmospheric drag on the transfer time and ΔV required for the low thrust

transfer.¹³ All chemical systems were assumed to burn impulsively.

Constant, Circumferential Thrusting

In operation both the Big and Little LEO satellite systems have active, relatively high power payloads which require power in shade and sunlight.⁸ Because the payload is usually not in use during satellite delivery and disposal, the power could be made available to the propulsion system. Thus, in this study, the EP systems described are assumed to operate from the solar arrays during the sunlit portions of the trajectory and from the batteries in the shadow portion. This use of payload battery power for electric propulsion has precedent with North-South stationkeeping using arcjet thrusters on geostationary spacecraft. It is assumed that the additional cycling and different charging patterns will have minimal impact on the multi-year power systems; a short electric propulsion orbit insertion and de-orbit adds only a few extra months to years of cycling.

One benefit of using the payload's power system in light and shade is the avoidance of non-thrusting periods during shadow. This should allow for simplified, circumferential steering. It can be shown with SEPSOT that the required in-plane steering angle for the sample big LEO spacecraft (see Big LEOS section) without shadowing is 0° or simply circumferential (i.e., perpendicular to the radius direction in the plane of the orbit) as shown in Figure 1. The shaded optimal steering is more complex and varies depending on shadow conditions throughout the trajectory (a sample for one orbit is shown in Figure 1). The corresponding ΔV is also higher for the non thrusting in shade case: 630 m/s versus 515 m/s for the constant thrusting case.

Circumferential steering simplifies the steering requirements on the spacecraft's guidance system. Using SEOR, this circumferential thrusting was tested assuming earth oblateness effects but neglecting atmospheric and solar drag effects. (The impacts of atmospheric and solar drag are assumed to be secondary.) The big LEO sample spacecraft (see Big LEOS section) reaches the targeted orbit with only a slight eccentricity; the perigee and apogee are only a few kilometers in error. Errors of this magnitude also occur for chemical stages¹⁰ and can be easily removed. Assuming the same propulsion system and circumferential steering but with shading, SEOR produces a significant eccentricity. In this case the perigee and apogee are in error by over 300 km as

shown in Figure 2. This orbit would have to be corrected with an almost 200 m/s ΔV and take on the order of two weeks using optimal steering from SEPSOT.

Other power/orbit/steering scenarios are possible. For instance, using all the available, beginning-of-life (BOL) solar array power, a higher power (but heavier) electric thruster system could be used but only during sunlit portions of the orbit. Such a trajectory would require more complex steering as shown above. In addition, the BOL power would not be available at the end-of-life and thus would require a throttleable thruster system. Another possibility would be to use shorter electric propulsion burns and start in an elliptical orbit; the electric propulsion system imitates a chemical thruster. This method, while reducing ΔV , would probably require a longer trip time as shown by Pollard and Janson.¹⁴ These options will be considered in further analyses.

SYSTEM ASSUMPTIONS

Several candidate propulsion systems were assumed in the analysis performed for this study. The candidate systems were meant to be representative and to show the benefits of a range of propulsion options. For the 1 kW class Big LEO example, the candidate electric propulsion systems were hydrazine arcjets, xenon Hall thrusters, and xenon ion thrusters shown in Table I.⁵ Each of the systems is either currently available or under development. More information concerning each can be found in the referenced texts. The 0.1 kW class Little LEO sample mission used ammonia resistojets,¹⁵ pulsed plasma thrusters (PPTs)¹⁶ and Hall thrusters¹⁷ shown in Table II. The ammonia resistojets and Hall thrusters assumed valve component miniaturization. State-of-art (SOA) hydrazine monopropellant thrusters were used as baselines for both sample missions. The Big LEOS sample mission assumed a 5 kg dry mass (less tanks), an 8% tankage, and an Isp of 235 seconds. The Little LEOS sample mission assumed a 223 second hydrazine system with a dry mass of 1.7 kg (less tanks) and a 8% tankage fraction.¹⁸

RESULTS

Big LEOS Example

The Globalstar system of eight planes of six satellites each at an altitude of 1414 km and 52° inclination was chosen as the sample Big LEOS communication system.² This Big LEOS system will provide mobile telecommunications service. The satellite is assumed to be approximately 450 kg at launch with a payload power of 1.2 kW.² A 7 year lifetime is assumed

including the requirement for end-of-life de-orbit. A hydrazine chemical propulsion system is baselined for Big LEOS sample. The Delta 7420 which is assumed to deliver four Big LEOS satellites, was used in this analysis.

Baseline Chemical Scenario

The Big LEOS sample system was assumed to use a hydrazine chemical system (235 s I_{sp} , 8% tankage fraction) for the orbit insertion and the de-orbit. The Delta 7420 was assumed to deliver four, 450 kg Big LEOS to a 185 km x 1414 km orbit (noted as 'chem' in Figure 3.). In all cases the combination of Delta dry second stage, adapter, dispenser, and reserve masses was assumed to be 1588 kg.¹⁹ The on-board chemical system performs an apogee burn to raise the perigee to 1414 km and circularize the orbit. Assuming impulsive burns the energy required for this maneuver was calculated to be 313 m/s. After the 7 year lifetime the Big LEOS must be de-orbited. Which disposal orbit is to be used was unknown but using a 500 km perigee based on NASA recommendations to limit orbit debris was a good minimum assumption; a lower perigee was possible but would require more fuel.²⁰ This 500 km perigee is set to limit the orbit life time to a reasonable level. The energy to lower the orbit perigee to 500 km is 226 m/s. Neglecting orbit maintenance requirements (which should be relatively smaller), the total ΔV required was 539 m/s. The chemical hydrazine system mass required to perform these maneuvers, assuming a 450 kg initial mass, was 107 kg. Thus, the non-propulsive spacecraft mass required for performing the Big LEOS mission was assumed to be just over 340 kg.

Electric Propulsion Scenario

The approximately 340 kg non-propulsive mass of the Big LEOS found in the Baseline Chemical Scenario was also assumed for the electric propulsion scenarios. The chemical orbit insertion system was replaced by an electric propulsion (EP) system. A 1.2 kW hydrazine arcjet, 1.2 kW Hall and a 1.2 kW ion propulsion system were considered (see Table 1). Because the payload power is assumed to be 1.2 kW in sunlight and shadow, the EP system was assumed to run off the solar arrays in sunlit portions of the trajectory and the batteries in the shadow portion. This use of payload battery power for electric propulsion was described in the mission analysis section. It was assumed that the additional cycling and different charging pattern will have minimal impact on the assumed 7 year system.

Instead of the elliptical Hohmann transfer target orbit of the chemical baseline mission, the EP Big LEOS would begin from a low circular orbit (Figure 3). Five EP Big LEOS will be launched to this low circular orbit. The EP system was tasked with raising the spacecraft to the final 1414 km circular orbit and de-orbiting the spacecraft. In keeping with the simplified tangential steering of the orbit insertion, a target 500 km circular disposal orbit was sought to fulfill the NASA recommendation. The energy required for the de-orbit is found to be 460 m/s.

The resulting mass breakdowns using each EP system are shown in Figure 4. By using a Hall thruster or Ion thruster the required EP circular starting orbits were 541 km, and 575 km with trip times of 28 and 31 days, respectively. Note that the higher thrust of the Hall system allows for a quicker trip time even though a larger orbit change is performed. De-orbit times were 29 and 34 days for Hall and ion thrusters, respectively. Spacecraft launch masses for each propulsion option are shown in Figure 5.

For the Hall and ion thrusters the higher starting orbits could be lowered to 400 km (to avoid excessive drag) and additional payload could be added to the five spacecraft but a sixth spacecraft could not be added. Alternatively, the life of the spacecraft could be extended by adding to the life-limiting parts of the bus (e.g., solar arrays and batteries). The spiral time and starting orbit could also be adjusted to help modify the final right ascension of the ascending node to the desired value.¹⁴

Lowering the starting orbit of the arcjet thrusters to 400 km did not allow for the additional spacecraft to be launched, but could allow for payload mass enhancement. The mass breakdown for the arcjet system is shown in Figure 4.

Packaging of an additional satellite into the Delta 7420 fairing was not considered in this analysis due to lack of packaging and dispenser information. However, assuming the body of the satellite is 1.8 x 1.5 x 0.6 m,² a bus volume for each Big LEOS satellite is 1.6 m³. The cylindrical portions of the Delta 2.9 m fairing have over 16 m³ of volume.⁹ Even allowing for array packaging and dispenser integration the addition of an extra satellite appears to be possible.

For this Big LEOs system, the total constellation of 48 satellites including 8 spares must be launched to provide complete service. Assuming all the satellites were to be launched on Deltas, fourteen launch vehicles would be required: 56 satellites / 4 per launch = 14 Deltas. With electric propulsion adding one satellite per launch almost three Delta launch vehicles could be saved: 56 satellites / 5 per launch = 11 Deltas plus one satellite. This eighth spare satellite could perhaps piggy back on another launch for a nominal fee.

Little LEOS Example

The Orbcomm system of three planes of eight satellites each at an altitude of 775 km and 45 inclination was chosen as the sample Little LEOS communication system. For this analysis each Little LEOS sample satellite weighs 40 kg at launch and was based on the enhanced microstar bus with an assumed constantly available payload power of 70 W using GaAs arrays and hydrazine chemical propulsion.²¹ A four year lifetime and an end-of-life de-orbit of the spacecraft is assumed. Launches are assumed to be eight at a time on a Pegasus XL launch vehicle.

Baseline Chemical Scenario

The assumed Little LEOS system uses an onboard hydrazine chemical system (223 s Isp, 8% tankage fraction) for the initial orbit spacing and the de-orbit. The Hydrazine Auxiliary Propulsion System (HAPS) equipped Pegasus XL was assumed to deliver eight, 40 kg Little LEOS to the 775 km circular operating orbit. The on-board chemical system must then perform the initial orbit spacing (11m/s) and the de-orbit. As with the Big LEOS example, a 500 km end-of-life perigee was assumed based on NASA recommendations.²⁰ The energy to lower the orbit perigee to 500 km was calculated to be 73.5 m/s. The total chemical ΔV required was 84.5 m/s and the corresponding fuel and tank mass is 1.6 kg. The chemical hydrazine system (less fuel and tanks) needed to perform these maneuvers was 1.7 kg.¹⁸ Thus, the non-propulsive spacecraft mass needed for the Little LEOS mission was assumed to be 36.7 kg.

Electric Propulsion Scenario

A 36.7 kg Little LEOS non-propulsive mass from the chemical scenario was assumed to be the required non-propulsive mass for the electric propulsion options. The on-board chemical propulsion system was replaced in turn by a 0.07 kW ammonia resistojet, a 0.07 kW Hall thruster and a 0.07 kW

PPT (see Table 2). Because the payload power was assumed to be 0.07kW in sunlight and shadow, the EP system was assumed to run off the solar arrays in sunlit portions of the trajectory and the batteries in the shadow portion. This was the same scenario used in the Big LEOS example. It was assumed that the additional cycling and different charging pattern would have minimal impact on the four year system.

Instead of eight chemical scenario Little LEOS delivered to the final 775 km operational orbit, nine EP Little LEOS were dropped off into a lower, 400 km circular orbit using a Pegasus XL launch vehicle without the HAPS.¹⁰ The higher I_{sp} of EP allowed for a propulsion system with much more available ΔV which, in turn, allowed for the launch of nine spacecraft instead of eight. The EP system was also tasked with performing the initial satellite spacing and de-orbiting the spacecraft. In keeping with the simple circumferential steering of the orbit insertion, a target 500 km circular disposal orbit was again assumed. The energy required for the de-orbit is found to be 147 m/s. The assumed LEO starting orbit is set to 400 km to minimize atmospheric drag.

The required EP mission wet mass breakdowns for the propulsion systems are shown in Figure 6. All three electric propulsion systems, ammonia resistojet, PPT and the Hall thruster could deliver the nine spacecraft as shown in Figure 7. The TOAD analyzer was used to ensure that worst case drag was small compared to the EP thrust level. The orbit insertion times were 3 days, 25 days and 83 days for the resistojet, Hall thruster and PPT, respectively. De-orbit times were 2 days, 19 days, and 63 days for the resistojet, Hall thruster and PPT, respectively. The resistojet would probably be the best choice given its performance and simplicity.

The additional Little LEOS per launcher would allow for an on-orbit spare for each plane, eliminating the need for a separate launch to replace a premature failure. Alternatively, a secondary payload could be placed on the launch vehicle. The elimination of the HAPS stage should allow for an additional 16.5 cm thick Little LEOS satellite.

CONCLUSIONS

It was shown that the mass of an additional satellite can be added to multiple Big and Little LEO spacecraft launches by using electric propulsion for orbit insertion, satellite positioning, and de-orbit. Orbit insertion and de-orbit times can be less than a month, in some cases days. A simple circumferential

steering method was assumed which relies on the payload's solar array and battery power and eliminates the more complex steering required when shading of the solar arrays must be considered. Ammonia resistojets, Hall, and PPT thrusters allowed for an additional satellite to be added to a little (tens of kilograms) low earth orbit satellite multiple launch. Hall, and Ion thrusters allowed for an additional satellite to be added to a big (hundreds of kilograms) low earth orbit satellite multiple launch. Arcjets were not able to add an additional big low earth orbit satellite but could enhance payload mass. These additional satellites can be used to reduce the number of launch vehicles required.

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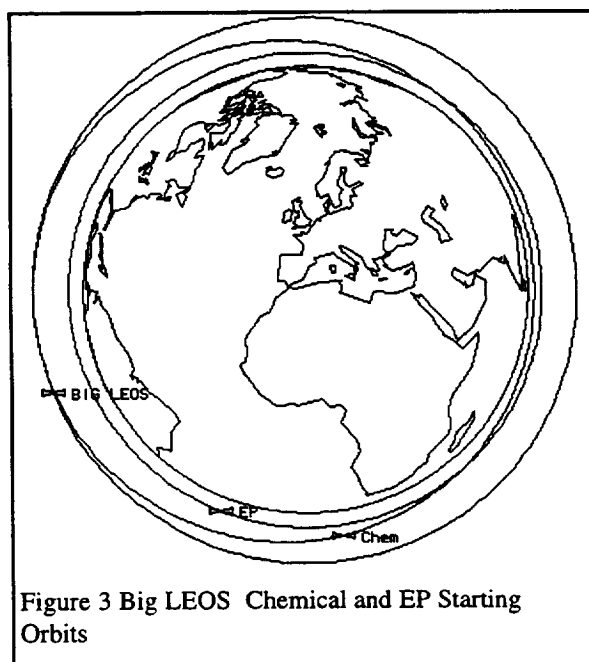
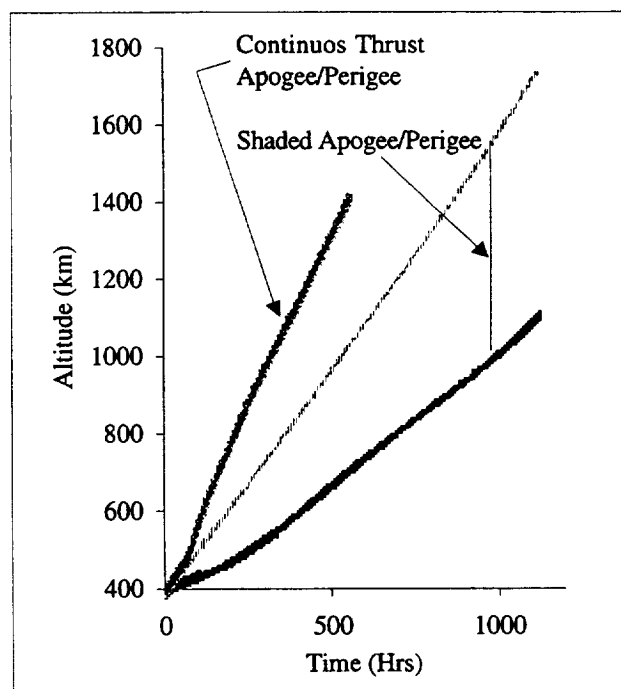
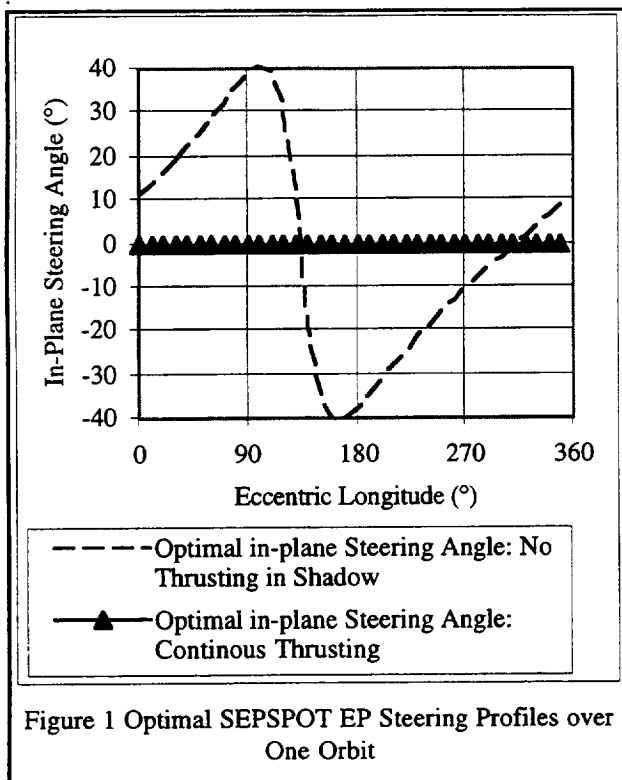
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Table I Candidate Electric Propulsion Systems for Big LEOS

Propulsion System Parameters	SOA N ₂ H ₄ Arcjet	Xenon Hall Thruster	Xenon Ion Thruster
Desired PPU Input Power Level	1.2 kW	1.2 kW	1.2 kW
Isp	585 s	1600 s	2500 s
Overall Efficiency (PPU & Thruster)	0.32	0.45	0.60
Tankage	7%	10%	10%
Masses:			
Thruster	1 kg	5 kg	7 kg
Gimbals	34 % of Thruster	34 % of Thruster	34 % of Thruster
Support	31% of Gimbals & Thrusters	31% of Gimbals & Thrusters	31% of Gimbals & Thrusters
Controller	0.55 kg/Thruster	0.55 kg/Thruster	1.55 kg/Thruster
Total Thruster + Gimbal + Support + Controller	2.3 kg/thruster	9.3 kg/thruster	13.8 kg/thruster
Feed System	0.8 kg/kWe	1.5 kg/kWe	1.5 kg/kWe
PPU	2.4 kg/kWe	4.7 kg/kWe	4.8 kg/kWe
Cabling	0.4 kg/kWe	0.4 kg/kWe	0.4 kg/kWe
Thermal Sys. (92% PPU)	31 kg/kWt-disp.	31 kg/kWt-disp.	31 kg/kWt-disp.
Total PPU + Feed + Cabling + Thermal	6.1 kg/kWe	9.1 kg/kWe	9.2 kg/kWe

Table II Candidate Electric Propulsion Systems for Little LEOS

Propulsion System Parameters	Ammonia Resistojet (RJ)	Xenon Hall Thruster	Pulsed Plasma Thruster
Desired PPU Input Power Level	70 W	70 W	70 W
Isp	300 s	1000 s	1228s
Overall Efficiency (PPU & Thruster)	0.7	0.28	0.10
Tankage	7%	20%	N/A
Total Thruster + Gimbal + Support + Feed System	0.59 kg/thruster	2.0 kg/thruster	4.5 kg/thruster complete (dry)
Total PPU + Feed + Cabling + Thermal	2.4 kg/kWe	15 kg/kWe	included in thruster



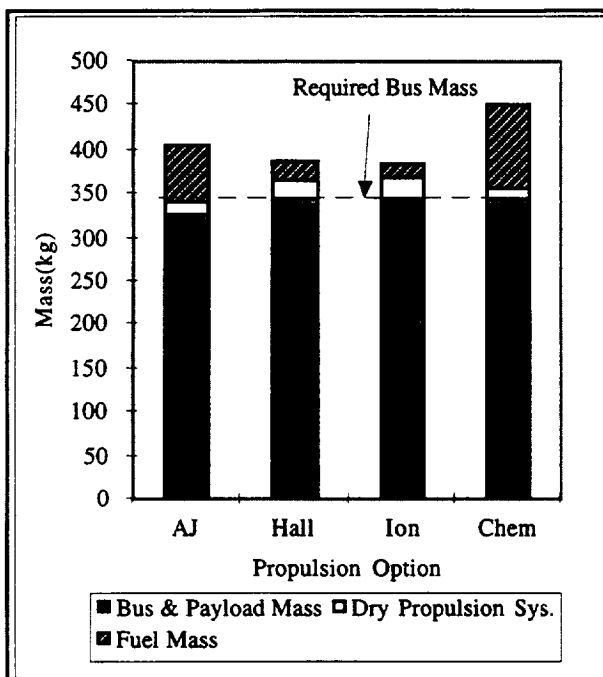


Figure 4 Big LEOS Masses

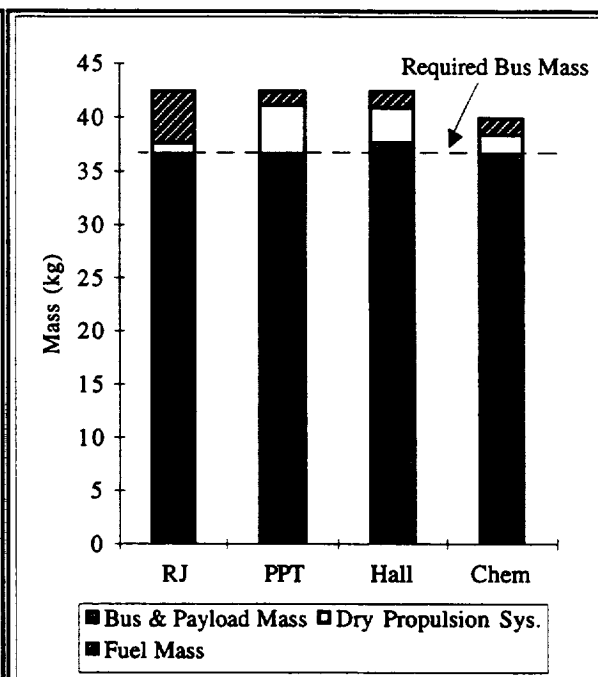


Figure 6 Little LEOS Masses

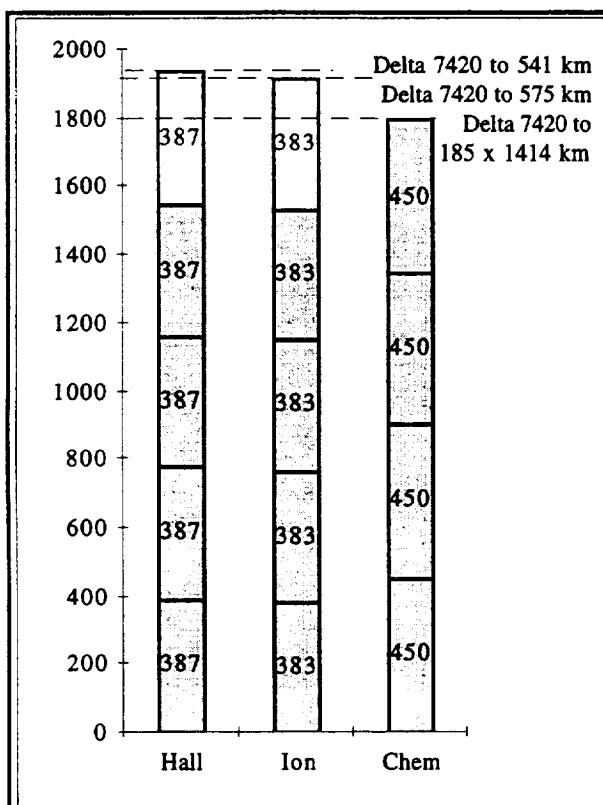


Figure 5 Big LEOS Launch Masses with Respect to Propulsion System

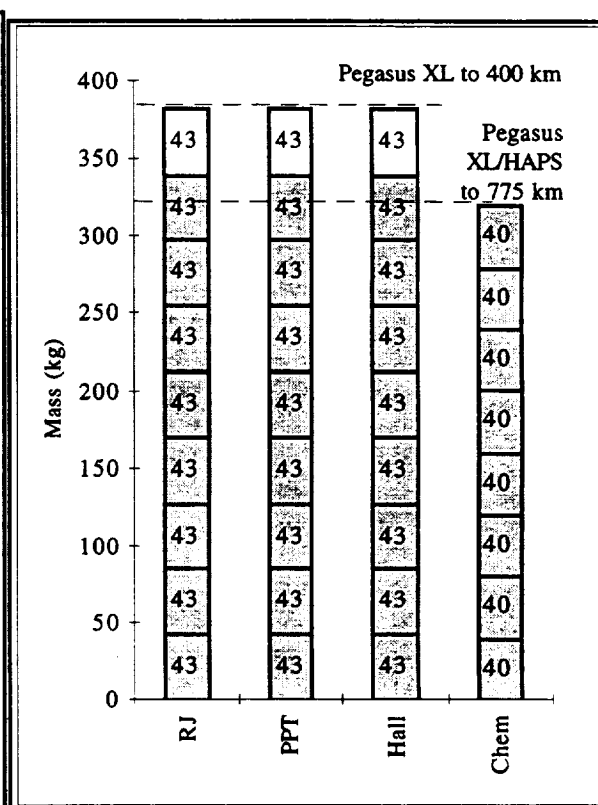


Figure 7 Little LEOS Launch Masses with Respect to Propulsion System

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